REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. REPORT DATE *(DD-MM-YYYY)*

2. REPORT TYPE
Technical Paper

3. DATES COVERED *(From - To)*
See Attached List

4. TITLE AND SUBTITLE
See Attached List

5a. CONTRACT NUMBER
N/A

5b. GRANT NUMBER
N/A

5c. PROGRAM ELEMENT NUMBER
N/A

6. AUTHOR(S)
See Attached List

5d. PROJECT NUMBER
N/A

5e. TASK NUMBER
N/A

5f. WORK UNIT NUMBER
N/A

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
See Attached List

8. PERFORMING ORGANIZATION REPORT NUMBER
N/A

9. SPONSORING / MONITORING AGENCY NAME(S) AND ADDRESS(ES)
Kristi Laug
AFRL/PROP
1950 Fifth Street
Wright-Patterson AFB OH 45433
937-255-3362

10. SPONSOR/MONITOR'S ACRONYM(S)
N/A

11. SPONSOR/MONITOR'S REPORT NUMBER(S)
N/A

12. DISTRIBUTION / AVAILABILITY STATEMENT

Distribution Statement A: Approved for public release; distribution is unlimited.

13. SUPPLEMENTARY NOTES
N/A

14. ABSTRACT

20030113 069

15. SUBJECT TERMS

16. SECURITY CLASSIFICATION OF: UNCLASSIFIED			17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON
a. REPORT	b. ABSTRACT	c. THIS PAGE	Unlimited Distribution	See Attached List	Kristi Laug 19b. TELEPHONE NUMBER *(include area code)* 937-255-3362

Dual Fuel Solar Thermal Propulsion for LEO to GEO Transfer: Ideal Rocket Analysis
AP.D10.006

AIAA 95-2840
Dual Fuel Solar Thermal Propulsion for LEO to GEO Transfer: Ideal Rocket Analysis

Jesse F. Stewart and
James A. Martin
The University of Alabama
Tuscaloosa, AL 35487-0280

31st AIAA/ASME/SAE/ASEE
Joint Propulsion Conference and Exhibit
July 10-12, 1995/San Diego, CA

DUAL FUEL SOLAR THERMAL PROPULSION FOR LEO TO GEO TRANSFER: IDEAL ROCKET ANALYSIS

Jesse F. Stewart*
James A. Martin**
University of Alabama
Tuscaloosa, AL 35487-0280

Abstract

Analysis of a dual fuel solar thermal propulsion concept was performed based on a system designed at NASA's Marshall Space Flight Center. The MSFC system uses a single fuel, hydrogen, to transfer 1,000 pounds of payload from LEO to GEO. Ammonia and hydrogen are used by the dual fuel system and both propellants were considered for use in the early stages of the mission. However, it was found that a system burning ammonia first was more suitable for the given mission. A fixed gross weight and the ideal rocket equation were used to calculate component weights. The analysis included some propellant losses. Payload weight was initially decreased by the addition of ammonia but it was increased by downsizing the power system to provide 2 pound of thrust with ammonia instead of with hydrogen. The analysis indicated that 1,000 pounds of payload could be placed into geosynchronous orbit with an ammonia fraction of about 14 percent of the gross weight. The tank volume was decreased by 20 percent and the propellant lost to boiloff was decreased by 24 percent. Also, thrust to weight variation with change in ammonia weight fraction was examined. Further analysis is required to fully weigh the benefits of a dual fuel solar thermal system.

Nomenclature

MSFC	Marshall Space Flight Center
LEO	Low Earth Orbit
GEO	Geosynchronous Equatorial Orbit
RCS	Reaction Control System
NH_3	Ammonia
H_2	Hydrogen
ΔV	Ideal velocity change
I_{sp}	Specific impulse of propellant
m_0	Gross mass of vehicle
m_{bo}	Burnout mass of vehicle
g_0	Gravitational acceleration at the Earth's surface
m_{1bo}	Mass after burn of first propellant
m_{20}	Mass at start of second propellant burn
m_{p1}	Mass of first propellant
m_{p2}	Mass of second propellant
m_f	Fixed mass of vehicle
m_{ps}	Mass of vehicle's power system
m_{t1}	Tank mass for first propellant
m_{t2}	Tank mass for second propellant
P	Power produced by thrust
T	Thrust produced by power system
$m_{\Delta v,prop}$	Propellant mass used to produce ΔV

* Undergraduate, Student Member AIAA
** Associate Professor of Aerospace Engineering,, Associate Fellow AIAA

Introduction

Solar thermal propulsion is a concept which makes use of the sun's energy to heat a working fluid as a means of providing thrust. The thrust is generated by expanding a superheated fluid through a nozzle. Note that although the term "burn" is used throughout this report, no combustion actually occurs. It is used to describe the use of propellant and is merely a convention established by the use of chemical rockets. The thrust level achieved depends on the temperature, properties, and exhaust velocity of the fluid. Because this thrust level is relatively low, the solar thermal concept can only be used as an upper stage to provide orbital transfers. Most of the solar thermal systems currently being studied use a single fuel. The most efficient fuel to date is hydrogen.[1] However, there are problems with these systems such as small payload volume and significant propellant losses. One possible solution to these problems is the development of a dual fuel solar thermal engine. This report details the initial stages of work on such a system. The dual fuel engine is designed to use ammonia and hydrogen during specific stages of the transfer mission, from low earth orbit to geosynchronous equatorial orbit, in an attempt to overcome the shortcomings of a hydrogen only system.

The work detailed in this report is based on a solar thermal engine system developed at the NASA Marshall Space Flight Center in Huntsville, Alabama.[1] Using the data from the MSFC analysis, the basic characteristics of the system were modeled. This system was then modified to act as a dual fuel system. The ideal rocket analysis was performed using a computer program. This program was used to find the weights of the components of the system. In addition data for a comparison of the thrust-to-weight ratio for each fuel before and after the switch point was produced using a separate computer program.

MSFC Single Fuel System

A solar thermal engine concept was designed at the NASA Marshall Space Flight Center by a team of engineers. The data contained herein summarizes the findings of their feasibility study as reported in March of 1994.[1] The MSFC system uses hydrogen, with a specific impulse of 860 seconds, to produce 2 pounds of thrust. It is designed to serve as an upper stage for a Lockheed LLV3 launch vehicle and provides an alternative to chemical upper stages. This alternative could deliver a greater payload weight for a given launch vehicle capability.

The system developed at MSFC (Figure 1a) consists of a single propellant tank, a solar energy collector system, and an absorber/thruster system. The collector system consists of two off axis parabolic mirrors mounted on a rotation and gimbal system to allow tracking of the sun as the spacecraft changes position. In order to minimize launch vehicle payload volume requirements, the collectors are inflated after the upper stage has separated. They are supported by a torus around the perimeter, and are connected to the rotation and gimbal system by rigidified inflated struts. The absorber system (Figure 1b) is made up of a windowless secondary concentrator leading to a blackbody absorber cavity encircled by fuel preheater tubes. The thrust generation process is illustrated in Figure 2. Sunlight passes from the collectors into the secondary concentrator. The concentrated solar energy then heats the blackbody walls of the absorber cavity. Heat from the absorber cavity passes through insulation and into the liquid hydrogen flowing through the tubes. The preheated fuel then flows into the absorber and is superheated. Thrust is created as high temperature hydorgen gas expands through the thruster nozzle. The attitude of the spacecraft is controlled by a reaction control system using a separate propellant supply.

Because of the low thrust level of a solar thermal stage, a direct transfer, such as a Hohmann transfer, cannot be used. Also, the typical low thrust transfer, a continuous burn spiral, cannot be used because the solar thermal stage requires the sun to generate thrust. If a spiral trajectory were attempted, the vehicle would pass behind the earth, the collectors would not be illuminated, and no thrust would be generated. The trajectory that is followed involves multiple propellant burns and is illustrated in Figure 3. To begin the orbital transfer, the vehicle increases its velocity by burning some propellant and moves from its circular orbit to an elliptical one. Because of the low thrust level, a ΔV large enough to place the stage onto an ellipse that touches GEO altitude is not possible, i.e. a Hohmann transfer is not possible. Thus, a number of burns are made at perigee, the point of closest approach, and the vehicle gradually brings its apogee altitude, the point farthest from Earth, to the destination altitude. The spacecraft then performs several burns at apogee to circularize its orbit.

The MSFC system was used as a baseline case for the development of a dual fuel system. The MSFC weight estimates and mission performance parameters were used to develop a computer program to compute the weights of the components of the rocket. The analysis was also based on the ideal rocket equation. In order to use the ideal velocity from the MSFC analysis, it was assumed that the thrust-to-weight history was matched. This allowed the correct burnout weight to be determined. Variations with changes in trust-to-weight history were not included in the analysis. Propellant losses were considered and were calculated based on the MSFC estimates of boiloff, leakage, startup, shutdown, and plume impingement. Plume impingement refers to the collision of ejected propellant with the portion of the spacecraft forward of the exhaust nozzle. The results of the ideal rocket analysis matched the MSFC results within ±3 percent when only hydrogen was used. Table 1 presents a summary of the component weights of the MSFC system, as determined by the computer program.

TABLE 1
Weight Summary for MSFC System

Tank Weight	405 lb
Thermal Control System Weight	97 lb
Tankage Weight	502 lb
Structure Weight	150 lb
RCS Weight	284 lb
Contingency Weight	266 lb
Fixed Weights	235 lb
Power System Weight	284 lb
Dry Weight	1721 lb
Payload Weight	983 lb
Burnout Weight	2704 lb
Hydrogen Weight (w/ losses)	2696 lb
Gross Weight	5400 lb

Ideal Rocket Equations

Because this work is only a preliminary step in the evaluation of the feasibility of solar thermal propulsion, the orbital transfer considered in this analysis is based on an ideal velocity approximation to the multiburn transfer from low earth orbit (LEO) to geosynchronous equatorial orbit (GEO). Only propellant losses due to boiloff, leakage, startup and shutdown, and plume impingement were considered. It should be noted that the procedure outlined below is based on mass. However, the actual analysis was performed using weights. It is possible to calculate the items below using weight because weight is directly proportional to mass, with the acceleration due to gravity at the Earth's surface as the proportionality constant. In other words, the weights calculated in the analysis are referenced to the surface of the Earth. Further study could consider such techniques as trajectory integration to further determine the practicality of a dual fuel solar thermal engine.

The basis for the analysis which follows is the ideal rocket equation.[2] For a single fuel rocket, it can be written as follows:

$$\Delta V = I_{sp} g_0 \ln \frac{m_0}{m_{bo}} \qquad (1)$$

where ΔV is the ideal velocity change required for the transfer, I_{sp} is the specific impulse of the fuel, g_0 is the acceleration due to gravity at the earth's surface, m_0 is the initial mass at LEO, and m_{bo} is the burnout mass at GEO.

For the engine considered in this analysis, two fuels are used in distinct stages of the transfer. Equation 1 must thus be modified to account for the different propellants:

$$\Delta V = \Delta V_1 + \Delta V_2 \qquad (2)$$

where ΔV_1 is the ideal velocity change produced by the first propellant and ΔV_2 is the ideal velocity change due to the use of the second. Equation 2 can be expressed in terms of specific impulses and masses by substituting equation 1 on the right hand side.

$$\Delta V = I_{s1} g_0 \ln \frac{m_0}{m_{1bo}} + I_{s2} g_0 \ln \frac{m_{20}}{m_{bo}} \qquad (3)$$

Here I_{s1} and I_{s2} represent the specific impulses of the first and second fuels respectively. Also, m_{1bo} is the mass of the spacecraft after the initial propellant has been expended, and m_{20} is the mass of the spacecraft at the initiation of the burning of the second propellant. In the ideal case m_{1bo} is equal to m_{20} because there are no losses in the instantaneous transition between fuels.

The total mass of the spacecraft, m_0, is the sum of component masses.

$$m_0 = m_{bo} + m_{p1} + m_{p2} \qquad (4)$$

Burnout mass, m_{bo}, can also be broken down into components.

$$m_{bo} = m_f + m_{ps} + m_{t1} + m_{t2} \qquad (5)$$

The component masses in equations 4 and 5 are defined as follows:

- m_{p1} is mass of first propellant
- m_{p2} is mass of second propellant
- m_f is fixed mass (communications systems, control systems, etc.)
- m_{ps} is mass of power system (thruster assembly, absorber, and collectors)
- m_{t1} is mass of tankage for first propellant
- m_{t2} is mass of tankage for second propellant

The power produced by the rocket can be determined from the following equation:

$$P = \tfrac{1}{2} T I_{sp} g_0 \qquad (6)$$

By placing these equations in a computer program, the baseline case from the MSFC data was approximated and component masses for a duel fuel system were computed.

Dual Fuel System

There are possible advantages to developing a dual fuel solar thermal engine. By using hydrogen and a heavier fuel such as ammonia, tank volume and tank weight can be decreased significantly from that required for a system using only hydrogen. This volume decrease allows the spacecraft to carry a larger payload. Propellant losses can also be decreased as a result of the use of a dual fuel system. For example, liquid hydrogen tends to boil off at a significant rate, as much as 5 percent of weight over a 30 day mission.[1] By using a propellant with a higher boiling point, in this case ammonia, in conjunction with the hydrogen, the propellant lost to boiling can be reduced. The long term goal of this analysis is to determine if a dual fuel system can significantly benefit a LEO to GEO transfer vehicle.

Several initial parameters were needed to begin the analysis; the following values are based upon those used by MSFC. The ideal ΔV for the transfer, from a LEO altitude of 400 nmi to GEO, was determined from the weight data from MSFC ($m_0 = 5{,}400$ lb, $m_{\Delta v, prop} = 2{,}140$ lb). Using the ideal rocket equation (Eqn. 1), the ideal ΔV was calculated as 13,964 ft/s. The effective specific impulse of hydrogen is 860 seconds, which produces 2 pounds of thrust in the system studied by Marshall Space Flight Center.[1] Ammonia's ideal specific impulse is 480 seconds, compared to 990 seconds for hydrogen. By taking a ratio of effective specific impulse to ideal specific impulse, the effective specific impulse of ammonia was calculated to be 417 seconds. A payload weight of 1,000 lb was calculated by MSFC's team of engineers. This was adopted as the target payload weight for this analysis.

Analysis of the ideal rocket was performed using a computer program written in Microsoft QuickBASIC. The inputs for the program were as follows: propellant 1 fraction (without losses) of m_0 (m_{p1}/m_0), initial weight (m_0), fixed weight, power system weight, specific impulse of propellant 1, specific impulse of propellant 2, total ideal ΔV, and a flag to indicate which propellant is used first, ammonia or hydrogen. The propellant fraction describes the amount of gross weight that is used to produce the ideal ΔV; it does not include any losses. Fixed weight includes the weights of the propulsion feed system; the electrical power system; the guidance, navigation, and control system; and the communication system. Values for these weights were taken from the MSFC feasibility report. Power system weight is made up of the weights of the absorber and the collectors, which were also assumed to be the same as the values given by MSFC. Using the ideal rocket equations, the following are calculated: ΔV due to propellant 1 burns, ΔV due to propellant 2 burns, burnout weight (m_{bo}), NH_3 propellant weight, NH_3 tank weight, H_2 propellant weight, H_2 tank weight, and payload weight. These weights are referenced to the surface of the Earth. The ΔV due to propellant 1 is obtained directly from Eqn. 1 with $m_{1bo} = (1 - m_{p1}/m_0)\, m_0$. The ΔV due to propellant 2 burns is calculated by subtracting the ΔV due to propellant 1 from the total ideal ΔV. This is then used to determine the final burnout weight at GEO by solving equation 1 for m_{bo} with m_0 equal to m_{1bo}. The weight of propellant 1 is $(m_{p1}/m_0)\, m_0$. The weight of propellant 2 can be found by subtracting burnout weight and propellant 1 weight from initial weight. Tank weights are calculated as a fraction of propellant weight. For Ammonia, tank weight is assumed to be 2 percent of propellant weight, and for hydrogen it is assumed to be 15 percent of propellant weight.[3] Notice that the tank weight per unit volume is nearly equal for the two propellants. Payload weight is the burnout weight, m_{bo}, minus the fixed weight, the power system weight, and the propellant tank weights.

The ideal rocket program was then modified to break the weights into the same components as those expressed in the MSFC report. Table 2 gives the weight breakdown and the relationships used to find the component weights. This was done to better facilitate comparison with the given data. All the weights and weight ratios are based on data given in the MSFC feasibility report, except those for ammonia tankage. The ammonia thermal control system percentage was calculated by direct proportion using the ratio of tank weight percentages (15/2). After providing a means of calculating component weights, the code was adapted to account for propellant losses due to boiloff, leakage, startup and shutdown, and plume impingement. Also, extra fuel needed for such factors as residuals, reserves, and absorber failure was included in the losses. The total loss of liquid hydrogen was assumed to be 26 percent of the weight of hydrogen. Five percent of this loss was assumed to be a result of boiloff.[1] Ammonia losses, except boiloff, were assumed to be the same as those for hydrogen. The boiling point of ammonia is much higher than that of liquid hydrogen. Therefore, the boiloff losses for ammonia can be neglected. The amount of ammonia lost is therefore only 21 percent. This estimate is conservative because ammonia, with its higher molecular weight and larger molecules, would not leak as fast as hydrogen. Using the above method, the Marshall Space Flight Center data was approximated within ±3 percent.

TABLE 2
Weight Relationships

Component	Mathematical Definition
Fixed Weight	$m_f = m_{pfs} + m_{eps} + m_{gnc} + m_{cs}$
• Propulsion Feed System	$m_{pfs} = 83$ lb
• Electrical Power System	$m_{eps} = 63$ lb
• Guidance, Navigation, Control	$m_{gnc} = 80$ lb
• Communications System	$m_{cs} = 9$ lb
Power System	$m_{ps} = m_{abs} + m_{col}$
• Absorber	$m_{abs} = 100$ lb
• Collectors	$m_{col} = 184$ lb
Hydrogen Tankage	$m_{t2} = m_{pt2} + m_{tcs2}$
• Propellant Tank	$m_{pt2} = .15\ m_{p2}$
• Thermal Control System	$m_{tcs2} = .036\ m_{p2}$
Ammonia Tankage	$m_{t1} = m_{pt1} + m_{tcs1}$
• Propellant Tank	$m_{pt1} = .02\ m_{p1}$
• Thermal Control System	$m_{tcs1} = .0048\ m_{p1}$
Tank Support Structure	$m_{tss} = .037\ (m_{p1} + m_{p2})$
Reaction Control System	$m_{rcs} = .03\ m_0$
RCS Propellant	$m_{rcsp} = .022\ m_0$
Contingency Weight	$m_{cont} = .2\ (m_{ps} + m_f + m_{t1} + m_{t2} + m_{rcs})$
Secondary Structure and Baffles	$m_{ssb} = .1(m_{pfs} + m_{abs} + m_{rcs} + m_{eps} + m_{gnc} + m_{cs})$

Initially, ammonia was burned during the early stages of the mission. Ammonia fraction (without losses) (m_{p1}/m_0) was varied from 0 to 40 percent in the analysis. The ammonia fraction is the portion of gross weight that is used to produce a velocity change. Figure 4 illustrates the change in propellant weight with ammonia weight fraction increase. Total propellant weight increases as a result of the increasing ammonia weight. There is some benefit from the addition of ammonia; the weight of liquid hydrogen decreases. Therefore, there will be less propellant lost to boiloff. Figure 5 shows variation of burnout weight, dry weight, and tank weight. This comes as a result of the decrease in burnout weight shown in Figure 5. All three weights decrease with increasing ammonia weight. The decrease in burnout weight results in a significant payload penalty. Figure 6 shows the payload weight change with increasing ammonia weight fraction. Figure 7 presents propellant volume variation with ammonia weight fraction. This figure illustrates that, because of the differences in density of ammonia and hydrogen, the volume of propellant decreases significantly. This decrease allows more space for payload.

Consideration was also given to burning hydrogen during the early stages of the mission. This could cut boiloff and leakage losses compared to a mission which uses ammonia first. The analysis was performed at hydrogen fractions (without losses) from 15 to 39.6 percent of gross weight. The upper limit corresponds to the hydrogen only system. The results of this analysis are shown graphically in Figure 8. Payload weight was graphed against ammonia weight fraction to allow comparison to the previous system, which burned ammonia first. Burning hydrogen first results in a greater payload weight decrease for the same increase in ammonia fraction. This indicates that the concept which burns ammonia first is more suitable for the given mission; the analyses that follow were performed on this configuration.

Reduced Power System

Increasing the ammonia fraction resulted in a decrease in payload weight. In a effort to offset this decrease, the power system, which consists of the collectors, the absorber, and the thruster, was downsized to provide 2 pounds of thrust with ammonia instead of with hydrogen. The results shown in Figures 4 through 7 are conservative because the thrust-to-weight would be increased, the ideal ΔV would be decreased, and the payload would be increased with a more accurate analysis. A more accurate analysis could include gravity and atmospheric drag losses and could involve consideration of the thrust-to-weight change. Equation 6 was used to perform the power system reduction. Using this formula, the amount of power required to produce the desired thrust level was computed for each propellant. The weights of the power system components were then adjusted by the ratio of power needed for ammonia to power needed for hydrogen. This ratio is equal to the ratio of specific impulses ($I_{sp,NH3}/I_{sp,H2} = 0.485$). The computer program used in the previous analysis was modified to downsize the appropriate weights.

Figures 9 and 10 illustrate the changes that resulted in dry weight and payload weight as a result of downsizing the power system. Dry weight decreases with respect to the original power system. Some of the payload weight reduction shown in Figure 6 can be regained by downscaling the power system. The resulting payload weight variation can be seen in Figure 10. For ammonia fractions up to about 0.16, there is a payload weight increase as well as a tank volume decrease, compared to the hydrogen only system. Burnout weight, tank weight, propellant weight, and propellant volume remained the same as those for the MSFC power system.

The results shown graphically above indicate that the target payload weight of 1,000 pounds can be achieved with an ammonia fraction of about 14 percent. Table 3 provides a weight summary for the dual fuel system. Compared the MSFC hydrogen only system, the total internal tank volume was reduced from about 610 cubic feet to about 485 cubic feet, a decrease of about 20 percent. This decrease in tank volume allows more space for payload. In addition, the total tank weight was reduced from about 500 pounds to about 400 pounds, once again a 20% decrease. This resulted in a 17 percent decrease in dry weight, from 1,720 pounds to 1,430 pounds. The losses from boiloff were reduced from 107 pounds to 81 pounds, a 24 percent decrease. The dual fuel system, burning ammonia in the early stages of the mission, is able to deliver the same payload as the MSFC system with a reduction in tank volume, tank weight, dry weight, and boiloff losses.

TABLE 3
Weight Summary for Dual Fuel System

Ammonia Tank Weight	18 lb
Thermal Control System Weight	5 lb
Ammonia Tankage Weight	23 lb
Hydrogen Tank Weight	308 lb
Thermal Control System Weight	74 lb
Hydrogen Tankage Weight	382 lb
Structure Weight	154 lb
RCS Weight	284 lb
Contingency Weight	218 lb
Fixed Weights	235 lb
Power System Weight	138 lb
Dry Weight	1434 lb
Payload Weight	1000 lb
Burnout Weight	2434 lb
Ammonia Weight (w/ losses)	915 lb
Hydrogen Weight (w/ losses)	2051 lb
Gross Weight	5400 lb

Also, the thrust-to-weight variation with changes in the ammonia weight fraction was considered. Downsizing the power system will reduce the thrust-to-weight after the fuel switch. This reduction will increase the ideal velocity. The results of this analysis are therefore optimistic. The governing equations in this portion of the analysis were the ideal rocket equation (Eqn. 1) and the power equation (Eqn. 6). The variation in thrust-to-weight ratio was computed using a computer code. Ammonia fractions from 0 to 0.4 were considered. Figure 11 shows the thrust-to-weight variance as propellant is burned and weight is decreased. The upper curve in the figure represents the thrust-to-weight for the MSFC system using only hydrogen. The bottom curve represents the downsized power system with an immediate switch to hydrogen, i.e. no ammonia is burned. For the dual fuel system, the thrust-to-weight variation follows the top curve until the fuel switch occurs. It then drops along one of the vertical lines to the bottom curve. Each vertical line represents a different fuel switch point, expressed in terms of ammonia fraction of gross weight. Figure 12 illustrates the change in ideal ΔV with decreasing weight. Note that for higher ammonia weight fractions, a larger ΔV is obtained at the propellant switch. The propellant switch can be seen as a change in the slope of the curves. The curve with a shallower slope corresponds to the burning of ammonia and the curves with steeper slopes correspond to the use of hydrogen. Figure 13 shows the thrust-to-weight variation with change in ΔV. The uppermost curve represents the burning of ammonia. At the fuel change, there is a step change to a lower curve. The center curve represents the thrust-to-weight variation of the MSFC system with hydrogen only. Note that, as a function of ΔV, the thrust-to-weight remains higher with higher ammonia fractions. The above analysis illustrates that using a dual fuel system. there are changes in the thrust-to-weight and ΔV history of the mission.

Future Considerations

Some further analyses are required to fully determine the relationship between the benefits and the costs of a dual fuel solar thermal upper stage. The most paramount of these considerations is a cost analysis. Also. losses from gravity, non-ideal burns, and atmospheric drag should be considered. The inclusion of these losses could give greater accuracy to the analysis. Trajectory integration would be a logical means of determining these losses. In addition, the propellant losses could be defined more precisely by determining the ammonia that is lost to leakage, startup/shutdown, and plume impingement. Consideration of the thrust-to-weight variation could also prove useful in the evaluation of a dual fuel system. Trade studies concerning different types of power systems could shed more light on the benefits of dual fuel solar thermal propulsion. This could include a variable geometry nozzle to make the most efficient use of both fuels. By taking these additional topics into account. the benefits and the costs of a dual fuel solar thermal system could be more accurately determined.

Conclusions

The results presented herein provide evidence that there are benefits to be gained from the use of a dual fuel configuration for a solar thermal orbital transfer vehicle. The addition of ammonia as the initial propellant caused a decrease in payload weight capacity despite the increase in available volume and decrease in tank weight. The increasing ammonia weight caused an the increase in the total propellant weight because the specific impulse of ammonia is significantly lower than that of hydrogen. The volume decrease, however, allows the payload to take up more space. When hydrogen is used during the early stages of the mission, the payload weight decreases more drastically than when ammonia is burned first. Therefore, it was concluded that the engine that uses ammonia first is more suitable for a LEO to GEO transfer. By downsizing the power system to give an ammonia thrust level of 2 pounds, the payload weight that was lost by addition of ammonia was regained. It was discovered that the target payload of 1,000 pounds could be achieved with an ammonia fraction of 14 percent. This fraction also resulted in decreased internal tank volume, tankage weight, dry weight, and boiloff losses. More space is available for payload as a result of the decreased tank volume. A more accurate analysis of the system with the inclusion of gravity, non-ideal burns, and atmospheric drag losses needs to be performed to completely define the benefits and the penalties of a dual fuel solar thermal system. Such an analysis could be accomplished using trajectory integration techniques. Also, a trade study concerning different types of power systems could shed more light on the benefits of dual fuel solar thermal propulsion.

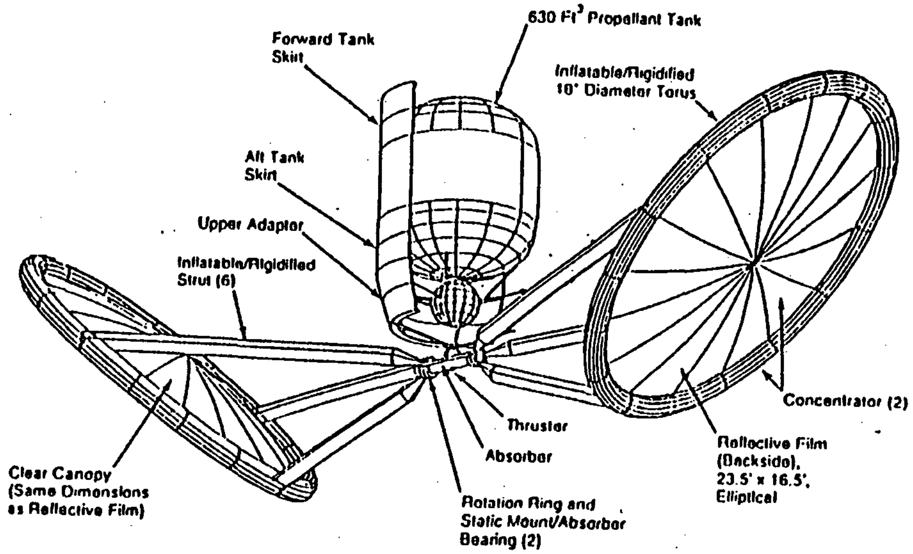
References

1. Patel, Saroj and William J. Emrich, Jr., *Solar Thermal Upper Stage (STUS) Feasibility Study*, NASA Marshall Space Flight Center, March 1994.

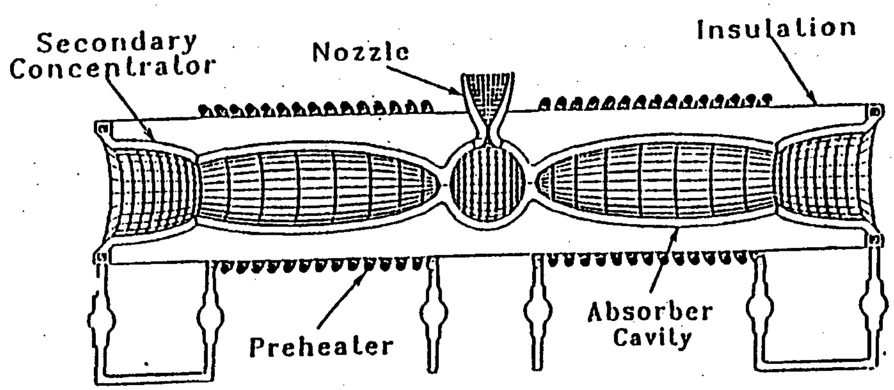
2. Sutton, George P., Rocket Propulsion Elements. An Introduction to the Engineering of Rockets, Sixth Edition, John Wiley & Sons, New York, NY, 1992.

3. Westerman, Kurt and R. F. Rochow, *Solar Hybrid Power and Propulsion System*, Babcock & Wilcox, Advanced Systems Engineering.

Figure 1
Marshall Space Flight Center
Solar Thermal Propulsion Concept



(a) Complete System



(b) Absorber/Thruster System

Figure 2
Process Illustration

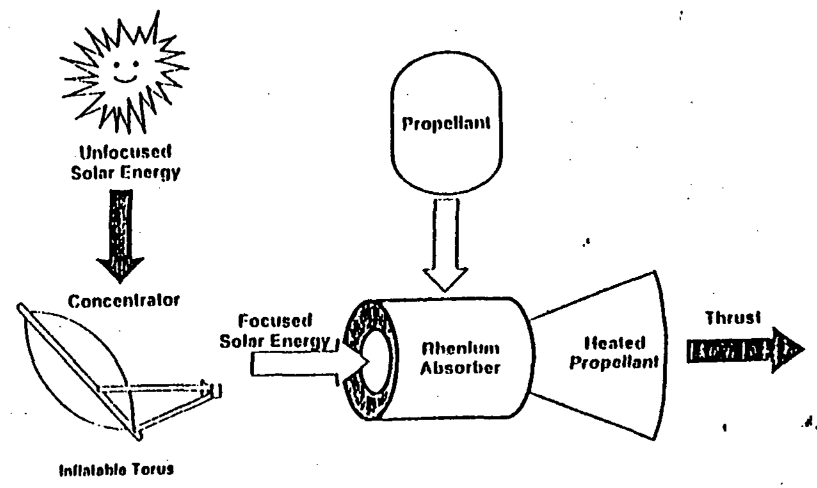
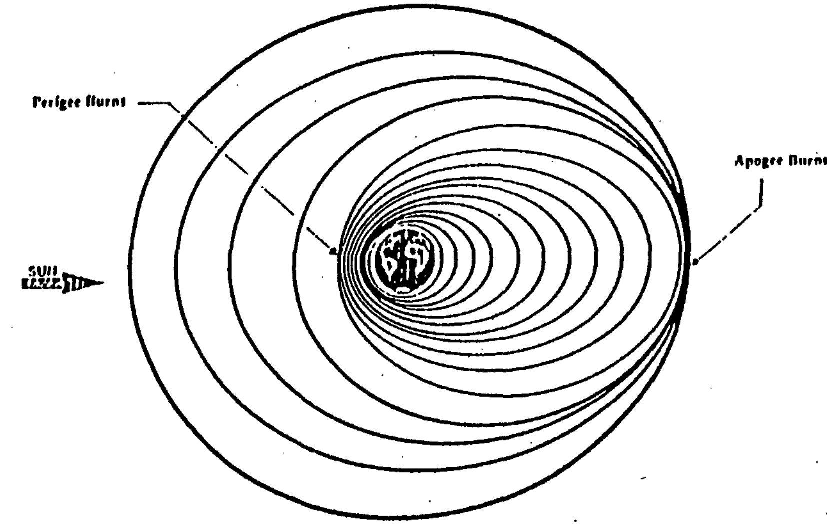


Figure 3
Mission Profile



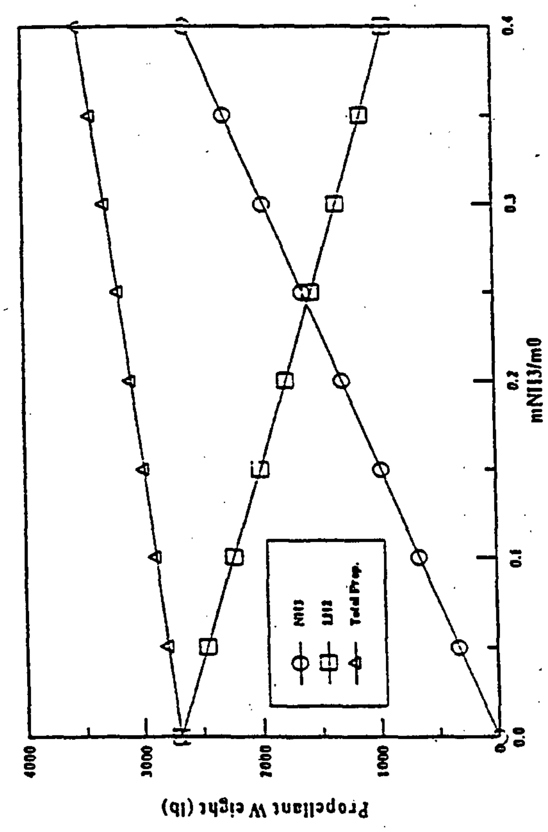


Figure 4
Propellant Weight Variation
with Increasing NH3 Weight

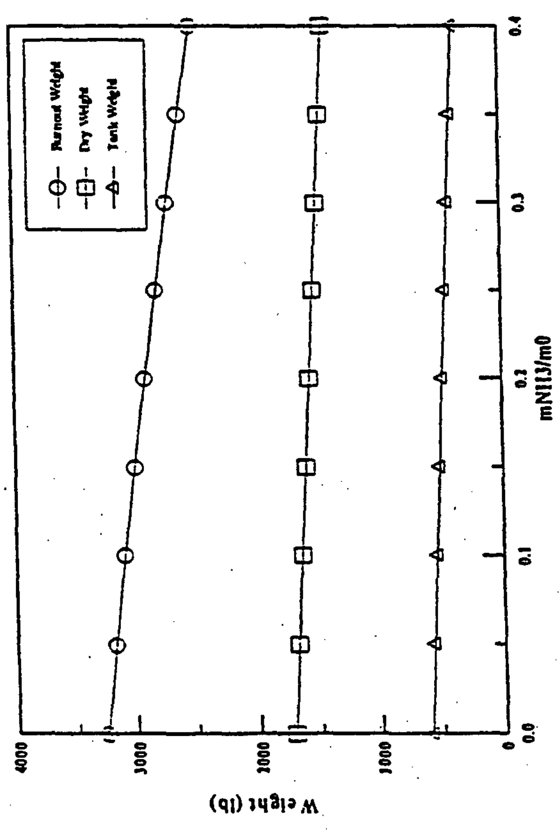


Figure 5
Weight Variation
with Increasing NH3 Weight

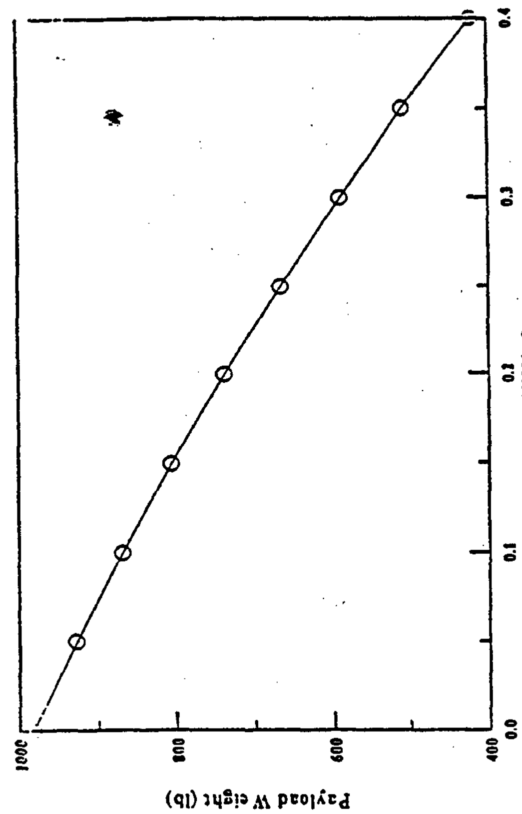


Figure 6
Payload Weight Variation
with Increasing NH3 Weight

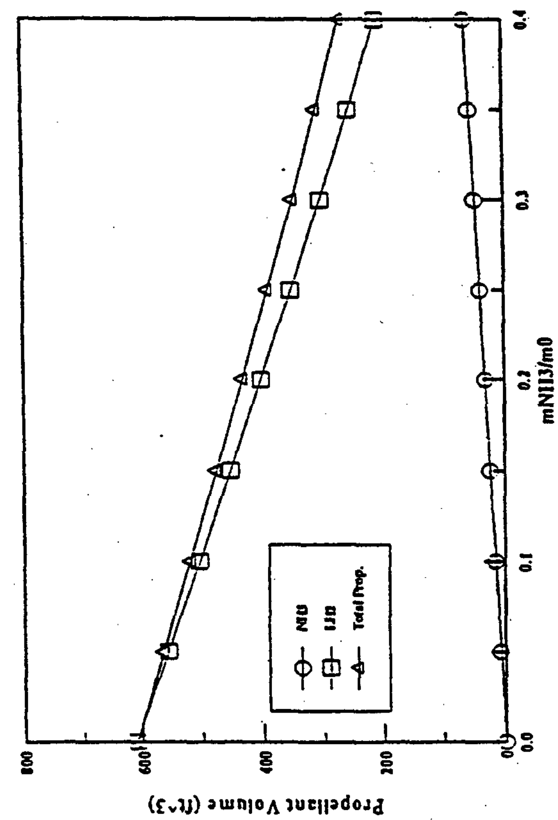


Figure 7
Propellant Volume Variation
with Increasing NH3 Weight

Figure 8
Payload Weight Variation
with Increasing NH3 Weight

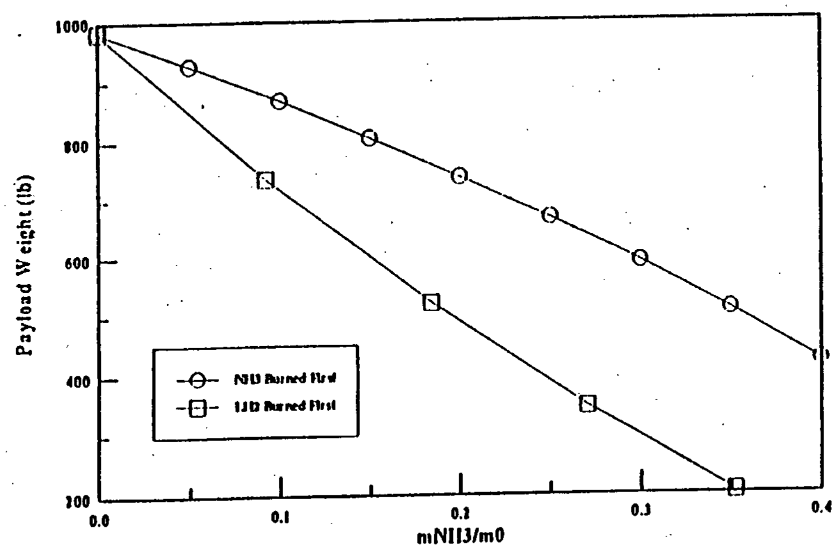


Figure 10
Payload Weight Variation
with Increasing NH3 Weight

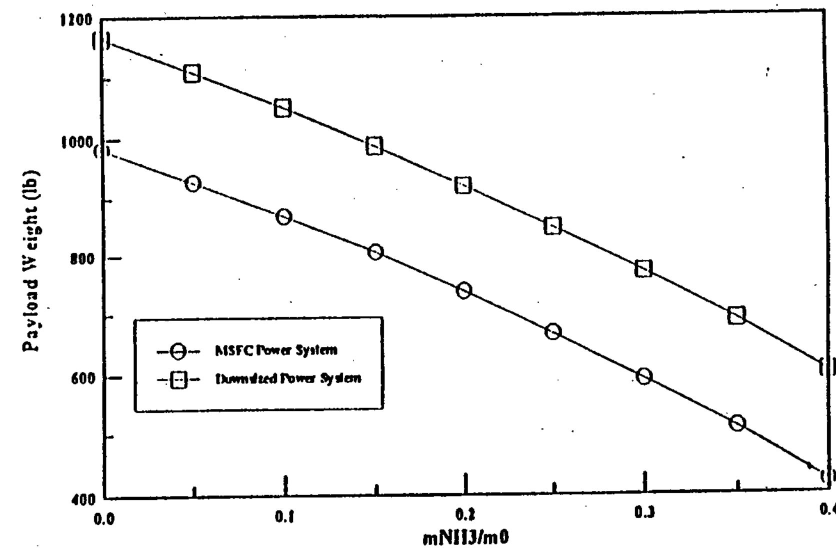


Figure 9
Dry Weight Variation
with Increasing NH3 Weight

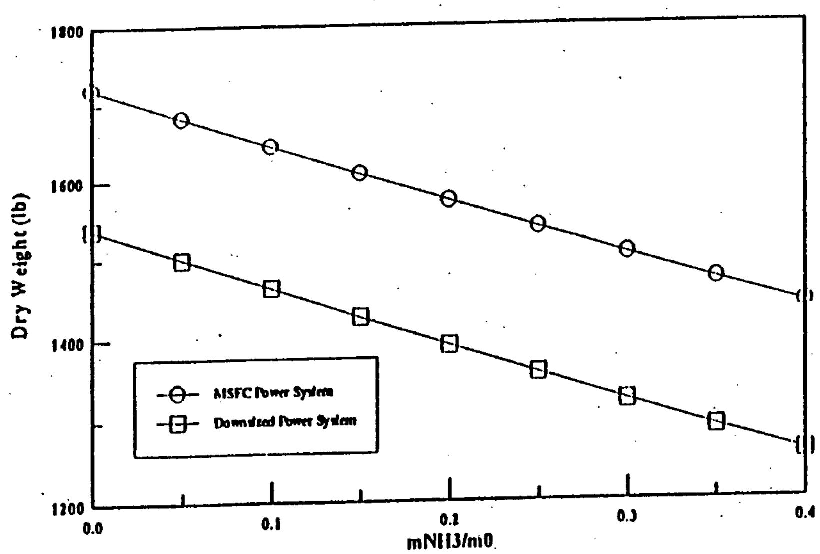


Figure 11
Thrust to Weight Variation
as Weight Decreases

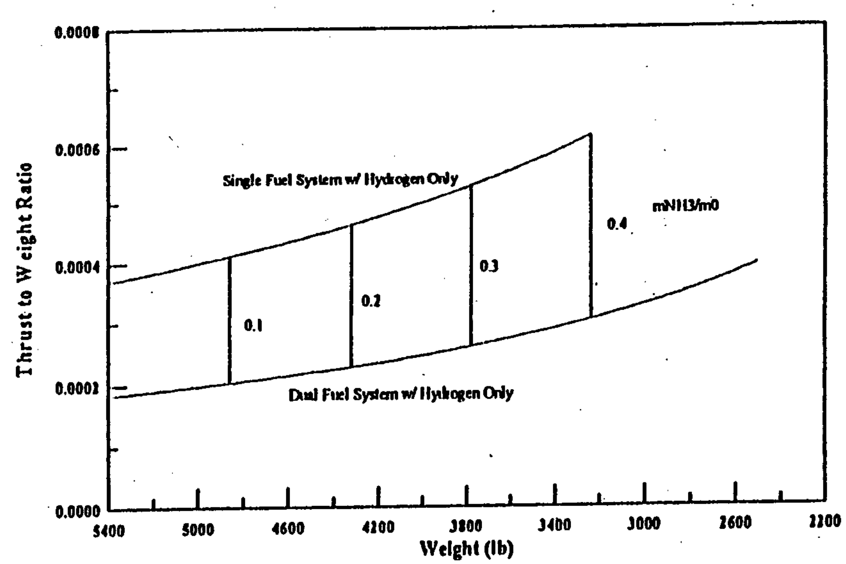


Figure 12
DeltaV Variation as Weight Decreases

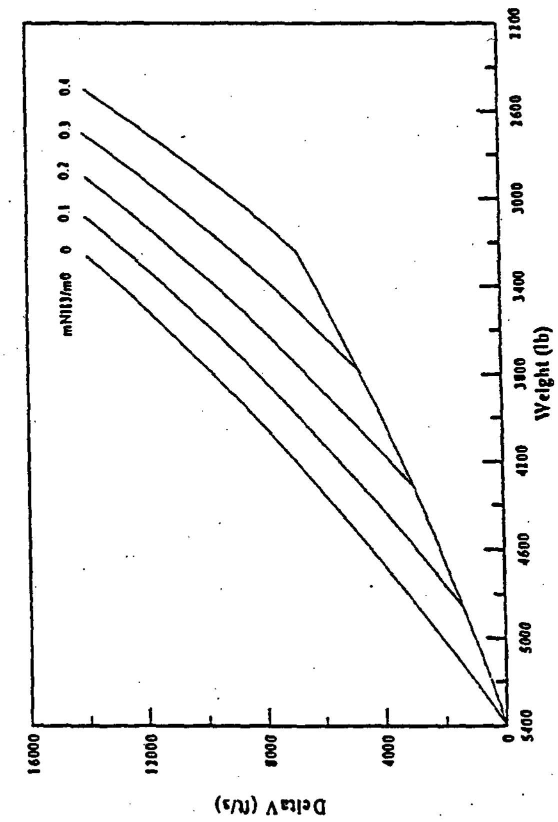


Figure 13
Thrust to Weight Variation with DeltaV